N91-22154

Vision-21, Space Travel for the Next Millenium
NASA Lewis Research Center, Cleveland, OH, April 3-4, 1990

COMPARISON OF SUPER-HIGH-ENERGY-PROPULSION-SYSTEMS BASED ON METALLIC HYDROGEN PROPELLANT FOR ES TO LEO SPACE TRANSPORTATION

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ABSTRACT

This paper investigates the application of metallic hydrogen as rocket propellant, which contains a specific energy of about 52 kcal/g in theory yielding a maximum specific impulse of 1700 s. With the convincing advantage of having a density fourteen times that of conventional liquid hydrogen/liquid oxygen propellants, metallic hydrogen could satisfy the demands of advanced launch vehicle propulsion for the next millenium.
Provided, that there is an atomic metallic state of hydrogen, and that this state will be metastable at ambient pressure, which is still not proven, the present publication shows the results of the investigation of some important problem areas, which concern the production of metallic hydrogen, the combustion, chamber cooling and storage.
The results show, that the use of metallic hydrogen as rocket propellant could lead to revolutionary changes in space vehicle philosophy towards small size, small weight and high performance SSTO systems. The use of high metallic hydrogen mass fractions results in a dramatic reduction of required propellant volume, while gas temperatures in the combustion chamber exceed 5000 K. Furthermore it follows from this study, that hydrogen (liquid or slush) is the most favourable candidate as working fluid. However, jet generated noise has to come into intense consideration, due to the very high exhaust velocities, which are possible with metallic hydrogen propellant.

Symbols and abbreviations:

Ce	: Exhaust velocity
DV	: Velocity increment
Hmet	: Metallic hydrogen
LHmet	: Liquid metallic hydrogen
SHmet	: Solid metallic hydrogen
MA	: Methyl alcohol
M0	: Overall launch mass
M1	: Payload mass
M8	: Propellant mass
Mn	: Vehicle dry mass
LH2	: Liquid hydrogen
SLH2	: Slush hydrogen
SH2	: Solid hydrogen
RP1	: Rocket propellant number one

CONTENT:

1. INTRODUCTION

During the last thirty years space technology became a more and more economic factor in many areas. In particular, in the fields of ES-LEO transportation systems but also in many fields of satellites applications the aspect of competitiveness gained in significance. Against the background of the expected increasing space activities in number during the next decades, assisted by the establishment of growing space stations as well as by expanded missions to other celestial bodies up to its colonization (moon and mars), the following aspects concerning the realization of those space projects should be regarded with priority:

a) Reduction of the specific transportation costs ($/kg-payload) by the factor 10 from today's level (about 25000 to 40000 $/kg)
b) Increase of space transportation capacities
c) Increase of space transportation vehicle performance
d) Reduction of negative implications to the environment (noise power, exhaust gas reactions with ambient air, required propellant mass per kg payload, etc.)
e) Increase of reliability

In view of these demands, the propulsion system represents the decisive influencing component of launchers. The performance of a rocket engine itself will be mainly influenced from the quality of the used propellant combination, which is characterized primarily by the specific impulse.

PROPELLANT COMBINATION	Isp (S)	DV/Ce	M1/M0	$/kg-M1	M8/M1
LOW ENERGETIC	<280	HIGH	VERY LOW	HIGH	HIGH
MEDIUM ENERG	280-330	MEDIUM	LOW	MEDIUM	MEDIUM
HIGH ENERGETIC	330-500	LOW	MEDIUM	LOW	LOW
SUPER HIGH ENERGETIC	>500	VERY LOW	HIGH	VERY LOW	VERY LOW

Isp: Specific impulse
DV/Ce: Propellant performance parameter
M1/M0: Payload mass ratio
$/kg-M1: Specific transportation cost
M8/M1: Specific propellant consumption

Tab.1-1: General correlation between propellant performance and main launch vehicle parameters

In Tab.1-1 the effects of the realizable specific impulse on the characteristic performance parameters of launch systems is shown. A diminishing value of the engine performance parameter DV/Ce (corresponding to challenge c) as well as increasing lightweight construction capability lead to increasing payload ratios (M1/M0) and therewith to increasing space transportation capacities (corresponding to challenge b). Increased payload mass ratios could lead to reduced specific transportation costs (corresponding to challenge a).

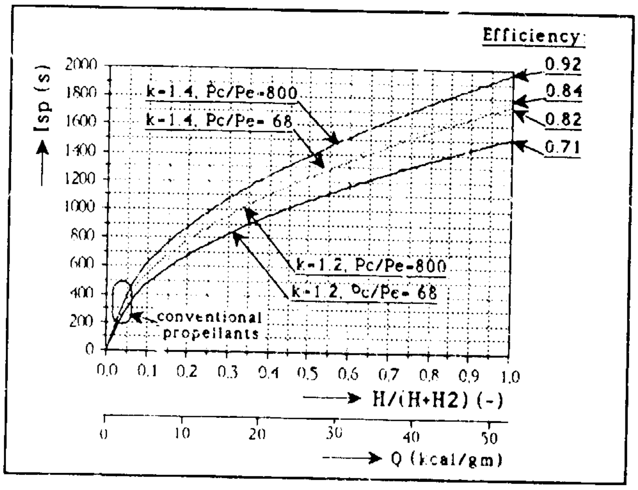


Fig. 1-1: Influence of weight fraction of free hydrogen radicals in the propellant, H/(H+H2), respectively the specific energy, Q, on the specific impulse, Isp, for various efficiency factors; k: ratio of specific heats, Pc: chamber pressure, Pe: nozzle exit pressure

As may be seen below, the use of super high energy propellants could reduce the propellant consumption considerably, in the course of which metallic hydrogen represents the most promising candidate. The interest of this study into metallic hydrogen as rocket propellant is based on the very high energy content, which is equivalent to high theoretical performance, which in turn is indicated by the specific impulse.
The specific impulse formula used, is as function of the overall net energy release per unit mass of propellant and is derived from the equation of the ideal exhaust velocity of a gas after thermodynamic expansion. Introducing an overall efficiency factor for the energy conversion (as a function of the ratio of specific heats, k, and pressure ratio, Pc/Pe) lead to:

$$Isp=(2\cdot h\cdot Q)^{0.5}/g \text{ (s)}$$

where

h: Efficiency factor for energy conversion $=1-(Pc/Pe)^{(k-1)/k}$
Q: Specific energy [J/kg] (= 4.184E6·Q[kcal/gm])
g: Gravitational constant (9.81 m/s²)

Using molecular hydrogen as working fluid, the energy release of the propellant can be calculated by multiplying the specific energy of metallic hydrogen by the weight fraction of the energized spezies in the propellant. For this case, complete free radical reactions and complete quenching of the metastable species to the ground state will be assumed.
In Fig. 1-1 the values for the specific impulse are plotted over the weight fraction of free hydrogen radicals in the propellant and the specific energy, considering various efficiency factors.
In the following chapters, the results of the investigation of some identificated problem areas concerning the application of metallic hydrogen as rocket propellant will be given, on condition, that atomic metallic hydrogen will exist and be metastable.

2. METALLIC HYDROGEN HISTORY

The interest in metallic hydrogen as rocket propellant results from the very high bonding energy between the atoms of the hydrogen molecule, combined with a very high density (probably 1150 kg/m3). The amount of energy which is necessary for the production of Hmet, that means the homolytic dissociation of molecular hydrogen into the metallic atomic state (H_2 <--> 2H), could be induced by pressure energy.

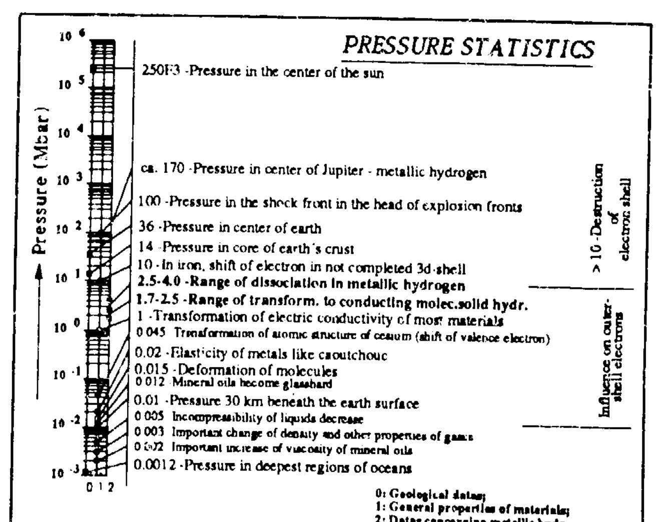


Fig.2-1: Pressure impacts

The corresponding development of the research on metallic hydrogen was not only aimed at the possible use as rocket propellant but also due to using the supposed superconducting characteristics. Fig.2-2 shows in a selfexplanatory way the essential acknowlegdments of the metallic hydrogen research efforts. There are two different procedures showing the growing stage of knowledge, first on an experimental basis, and second on theoretical investigations. As may be seen, the main research

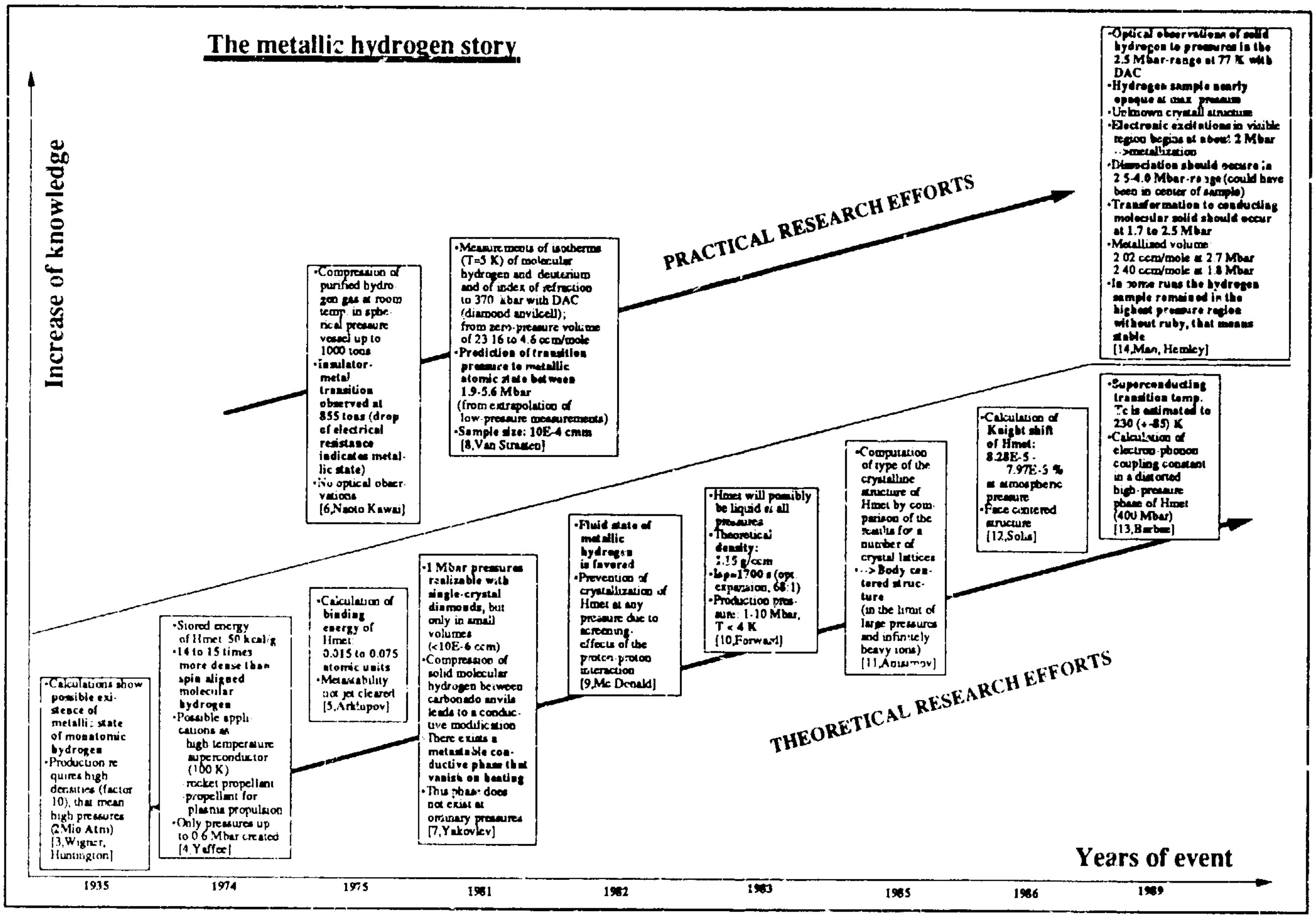


Fig.2-1: Development of research efforts in the field of metallic hydrogen over the years

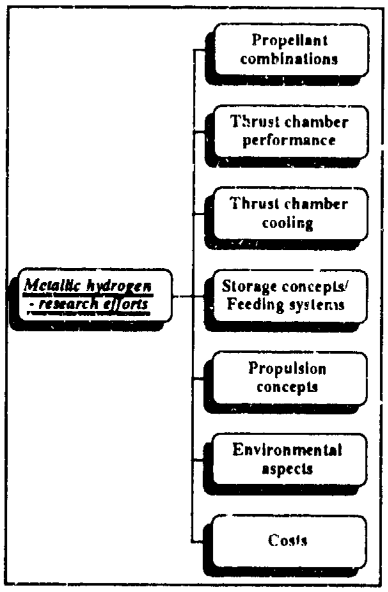
constraints concerned proving the existence of Hmet. The experimental investigations are very closely connected to high pressure research. To give an impression of the very high pressures of 2.5 to 4 Mbar, which will be required for Hmet production, Fig.2-1 illustrates different impacts due to the appearance of pressure.

The main properties of Hmet are summarized on Tab.2-1. For this study only those properties of metallic hydrogen are of interest, which are of importance for use as rocket propellant.

State of aggregation	liquid [9,10]	solid [13,14]	
Stored energy (100%)	52.1 kcal/g [4]		
Density	1.15 g/cm3 [10]		
Storage temperature	5 K [8]	77 K [14]	room temp. [6]
Storage pressure	2.5-4 Mbar [14]	ambient pressure	
Specific impulse	1679 s (Pc/Pe=68, k=1.3)		
Lattice structure	bcc [11]	fcc [12]	
Knight shift	8.28E-5 - 7.97E-5 % [12]		
Superconduction transition temperature	230 + 85 K [13]		
Optical property	opaque [10]	not opaque [14]	
Dissociation pressure	2.5-4 Mbar [14]		
Transformation pressure to conducting molecular solid	1.7-2.5 Mbar [14]		
Metastability	yes [14]	no [7]	

Tab.2-1: Properties of metallic hydrogen; properties important for this study are marked

3. MAIN RESEARCH EFFORTS



The present investigations will give a general idea of metallic hydrogen propellant application. Due to the wide range of investigation areas, only those will be presented, which have been identified as the most interesting ones. Fig. 3-1 illustrates the main research efforts.

Fig. 3-1: Main research efforts

3.1 CHOICE OF PROPELLANT COMBINATIONS

Due to the very high energetics of metallic hydrogen, yielding very high thrust chamber gas temperatures (as may be seen later), a secondary propellant component is required, which absorbs the heat of reaction and serves as expansion medium. The choice of the right working fluid should be done against the background of the general known propellant features.
Besides the combination of Hmet/LH2 as presented in the introduction, the following investigations include also more dense elements like slush hydrogen SLH2 and solid hydrogen SH2 which are cryogenic fuels, but also storable ones like the conventional used rocket propellant RP1 and another candidate, namely methyl alcohol, which offers higher performances compared to RP1. Tab. 3-1 lists the basic data of the investigated reactants fuels.

Chemical	Formula	Enthalpy (cal/mole)	Phase	Temp.(K)	Density (g/cm^3)	Designation
Hydrogen	H2	-2154	liquid	20,27	0,0709	LH2
RP-1	C1H1.9423	-4530	liquid	298,15	0,773	RP1
Methyl alcohol	C1H4	-57040	liquid	298,15	0,7866	MA

Tab.3-1: Reactants fuels data

The choice of the propellant combinations determines the tankage concept. For lack of data about metallic hydrogen properties (will it be a liquid or a solid?, storage temperature?, etc. as explained in chapter 3) various possible propellant combinations alternatives have been regarded. They are listed in Tab.3-2 (where -P stands for powder, -G stands for grain). The marked propellant combinations (PC-x) will be analysed below.

	LHmet	LH2	RP1	MA	SLH2	SH2	SHmet-P	SHmet-G
LHmet		PC1	PC2	PC3	PC4			
LH2							PC5	PC6
SLH2							PC7	PC8
RP1							PC9	PC10
MA							PC11	PC12
SH2							PC13	PC14

PC: Propellant combination
PC1...4: Liquid systems
PC5...12: Hybrid systems
PC13,14: Solid systems

Tab.3-2: Investigated propellant combinations

3.2 THRUST CHAMBER PERFORMANCE PARAMETERS

The thrust chamber is the basic element of a chemical rocket engine. Typical chamber parameters (defined in Tab.3-3) are shown in the following figures 3-2 to 3-6, based on thermochemical computations which equate the heat of reaction of the propellant combinations and the rise in enthalpy of the combustion gases at frozen composition [16].

Parameter	Symbol	Unit
Specific heat ratio	κ	-
Characteristic velocity	C^*	m/s
Gas temperature	T_c	^oK
Bulk density	ϕ_b	kg/m^3
Vac. specific impulse	I_{vac}	m/s
Densitiy impulse	I_d	Ns/dm^3

Tab.3-3: Thrust chamber performance parameters

All parameters are plotted against the metallic hydrogen mass fraction (respectively weight fraction). All data are given for 68:1 expansion ratio. The energy storing capacity of the propellant gas molecules is indicated by the specific heat ratio. Increasing values of κ indicate decreasing energy storage capabilities, due to a lower number of degrees of freedoms, and in turn gives lower engine performance.

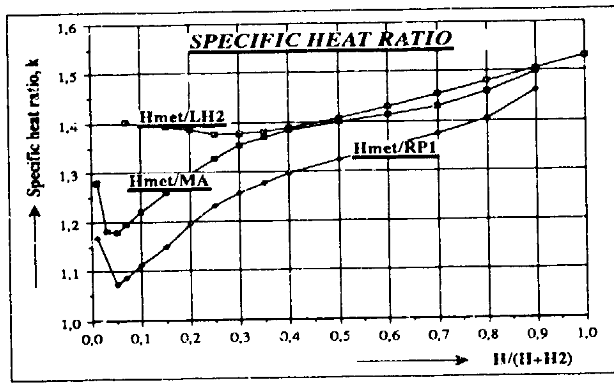


Fig.3-2: Theoretical specific heat ratio data for given propellant combinations (frozen flow, 68:1, Pc=50 bar)

Increasing atomic hydrogen weight fractions in the propellant yield higher thrust chamber gas temperatures and thus, will lead to higher atomic hydrogen mol fractions of the combustion gases. Simple atoms, however, have only three translational degrees of freedom, and hence, yield lower specific heats, which results in increasing values for κ.

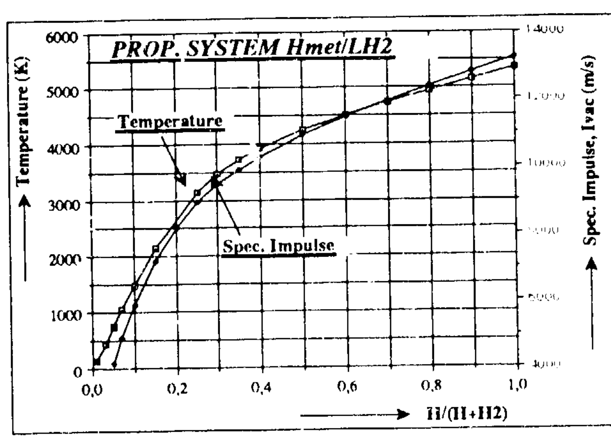


Fig.3-3: Theoretical data of gas temperature and vac. specific impulse for Hmet/LH2 combination (frozen flow, 68:1, Pc=50)

As may be seen from figures 3-3,4, the vacuum specific impulses of the systems Hmet/RP1 increases nearly linear for a wide range of Hmet weight fractions (nearly the same for system Hmet/MA), compared to the system Hmet/LH2. Moreover a very high temperature level is reached very soon with increasing Hmet weight fraction, compared to the system Hmet/LH2, which yields a more constant gradient for the temperature curve. Increased chamber pressures shift the curves to higher values for all combinations.

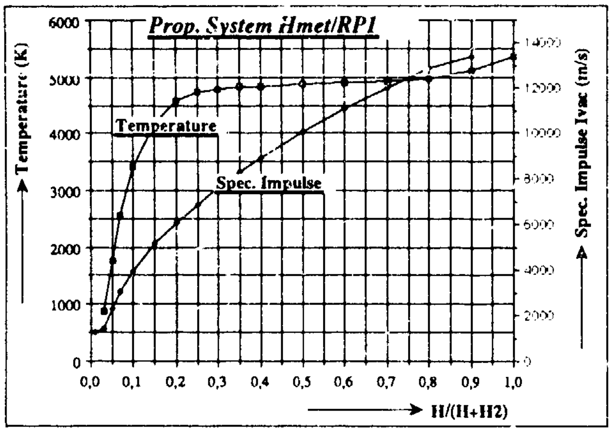


Fig.3-4: Theoretical data of gas temperature and vac. specific impulse for Hmet/RP1 combination (frozen flow, 68:1, Pc=50)

Results:

- •Hmet propulsion systems with specific impulses of today's values (4500 m/s), have combustion gas temperatures much lower (beneath 2000 K).
- •High Hmet weight fractions yield high specific impulses, but also high combustion gas temperatures (up to 5500 K)
- •The gas temperatures of the systems Hmet/RP1,MA rise rapidly with Hmet weight fraction, those of System Hmet/LH2 rise slowly.
- •An increase of %Hmet from 30% to 80% will hold gas temperatures nearly constant below 5000 K for systems Hmet/RP1,MA. The specific impulse increases constantly from about 7600/7000 m/s to 12500/12200 m/s within the same limits.
- •The specific impules of the system Hmet/LH2 rise more rapidly with Hmet weight fraction than those of the systems Hmet/RP1,MA.
- •The curves for temperature and Ivac of the system Hmet/H2 behave nearly identical.
- •Hydrogen as working fluid is advantageous, due to less temperature criticality.

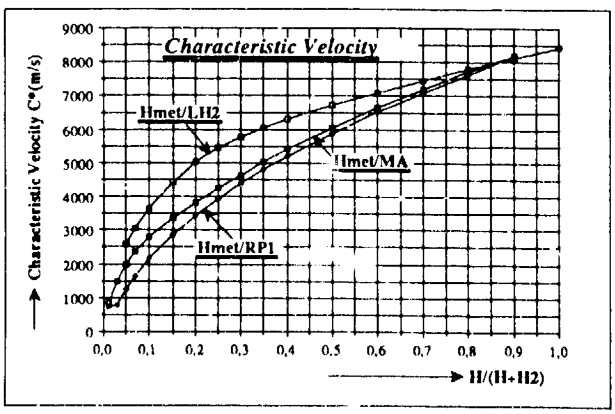


Fig.3-5: Theoretical data of characteristic velocity and vac. specific impulse for given propellant combinations (frozen flow, 68:1, Pc=50 bar)

The computations of the characteristic velocities yield the following results:

- •The higher values of the characteristic velocity in the sequence H2, MA, RP1 indicate a combustion process of higher energy and efficiency corresponding to a lower value of propellant consumption.
- •Hydrogen as working fluid represents the best alternative due to highest values for both, characteristic velocity and vac. specific impulse, at low Hmet weight fractions.

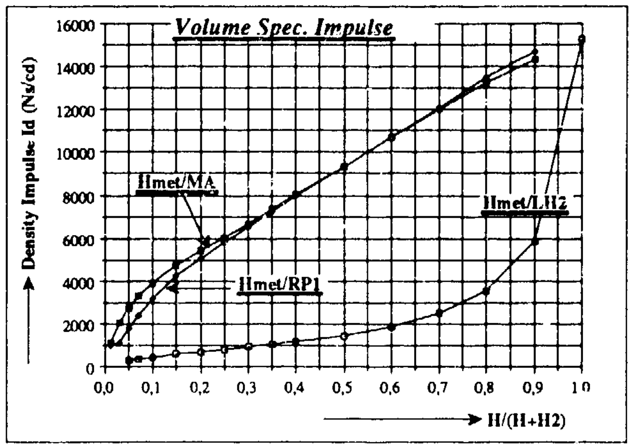


Fig.3-6: Theoretical data of density impulse for given propellant combinations (frozen flow, 68:1, Pc=50 bar)

The results of the computation of the volume specific impulse are:

- •The more dense working fluids RP1 and MA shows an impressive advantage concerning tank volume reductions, compared to hydrogen, which yields no more than 430 kg/m3 of bulk density even with 90% Hmet.
- •Therfore the density impulses are much higher, using RP1 or MA as working fluids.

To give a comparison of the combustion behaviors between the metallic hydrogen propellant combinations and other conventional liquid propellant combinations used today, the vac. specific impulse and gas temperature are plotted in Fig.3-7.
If the specific impulse with metallic hydrogen combustion is fixed on the level of the conventional high energetic combinations (LOX/LH2, LOX/F2), the gas temperatures will be lower. In the case of the constant gas temperature, the specific impulses are much higher.

Results:

- •There are lower thermal risks of thrust chamber due to lower chamber temperatures, if metallic hydrogen propulsion is used in the realm of conventional specific impulses (low percentages of metallic hydrogen).
- •An enormous increase in specific impulse arises, if chamber temperatures are not kept within conventional limits.
- •The combination using hydrogen as working fluid shows the most extreme behavior in this sense.

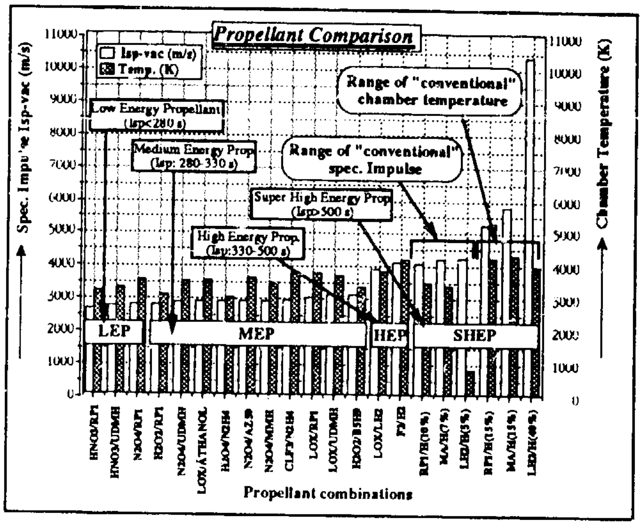


Fig.3-8: Comparison of various propellant combinations (frozen flow, 68:1, Pc=50 bar)

3.3. THRUST CHAMBER COOLING

In the case of increase of the metallic hydrogen fraction above 15% for the system Hmet/MA and Hmet/RP1 respectively above 40% for Hmet/LH2 the chamber temperatures will increase over the values for the conventional propellant combinations which lies in the range between 2500 K and 3700 K. Because of the high heat transfer rates from the hot gases to the chamber wall, thrust chamber cooling becomes a major design consideration. The objective was to investigate the influences of gas temperatures, arising from Hmet-combustion, on chamber cooling demands. The results can only be regarded as simple approximations.

Fig.3-9 shows schematically the cooling problem, which is basically one of heat and mass transport associated with conduction through a wall. It can be treated as a series type heat-transfer problem.

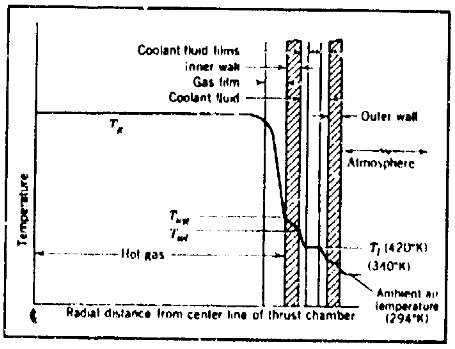


Fig.3-9: Temperature gradients through chamber wall [18])

The general steady-state heat-transfer equation can be expressed as follows:

$$q=H(T_g-T_l)=Q/A$$

where

- q : Heat flux or heat transferred per unit area per unit time [W/m2]
- H : Overall film coefficient or overall heat transfer coefficient [W/m² °K]
- Tg : Absolute chamber gas temperature [K]
- Tl : Absolute coolant liquid temperature [K]
- Q : Heat transfetred per unit time [W] across a surface A [m²]

The determination of the overall film coefficient H is a rather complex problem. It can be expressed as follows:

$$H=1/(1/h_g+t/k+1/h_c)$$

where

- t : Chamber wall thickness [m]
- k : Thermal conductivity of chamber wall [W/m °K]
- hg : Gas film coefficient [W/m² °K]
- hc : Coolant liquid film coefficient [W/m² °K]

H is composed of the individual coefficients for the boundary layers and the chamber wall. The smaller H, the smaller is q. It is one of the major design goals to keep gas side heat transfer coefficient hg low, but the coolant liquid heat transfer coefficient and conductivity t/k high, in relation to hg. The cooling problem will be analysed in a very simple manner, based on the given data for q and H of the SSME thrust chamber. It will simply be answered, how much the cooling parameters q and H will change relative to the SSME data, as function of the relative change of chamber temperature (which is in turn dependent on Hmet weight fraction).

The following table 3-4 states the most important material parameters exemplary for the SSME, which uses the today's most developed integral CuAgZr design.

Parameter	Unit	Integral CuAgZr design of SSME	Tendency
Thermal conductivity	W/cm K	3.3	should be high
Coefficient of themal expansion	K·1E-6	16.5	should be low
Wall thickness	cm	0.07	should be low
Overall film coefficient H	W/m² °K	8.797	should be high
Poissons ratio of inner shell material	-	0.34	should be high
Max. heat flux q	W/cm²	28300 (for e_{therm}=75%)	should be high

Tab.3-4: SSME chamber material parameters and overall tendency

Fig.3-10,11 gives an impression of the cooling difficulties induced by chamber temperatures above today's levels. The percent changes of the heat flux, dq, the temperature drops from absolute chamber gas temperature to the absolute liquid coolant temperature, dT, and the change of the overall film coefficient, dH, are plotted over the change of metallic hydrogen weight fraction for the combinations Hmet/LH2 (Fig.3-10) and Hmet/MA (Fig.3-11).

The zero-line represents the SSME technology with chamber temperature of Tc=3637 K and coolant liquid temperature of Tl=420 K. The changes for dq have been computed for constant dH and just the other way round, based on the general steady-heat tranfer equation. It should be noted here, that this approxi-

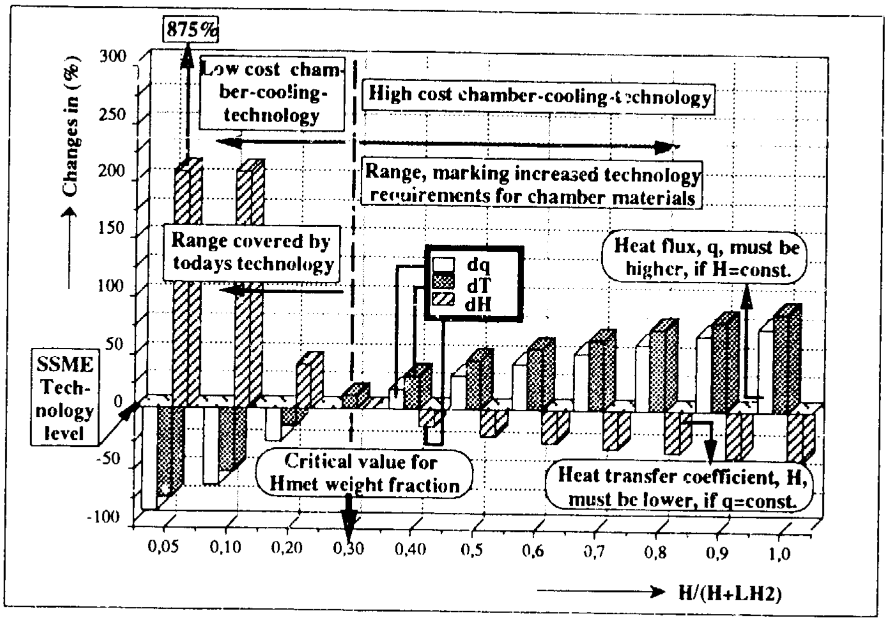


Fig.3-10: Sensitivity of chamber cooling for propellant system Hmet/LH2, relative to SSME chamber cooling

mat computation does not regard the influence of changed gas composition by using other propellants than in SSME. The results given below are optimistic limits. The points in the figures 3-10,3-11, where the bars cross zero, will probably move to the left, due to the fact, that the investigated propellants yield lower molecular weights than LOX/LH2, and hence yield higher gas side heat transfer rates.

The changes of premises indicate the beginning of the range, marking increased technology requirements for the chamber materials (at the point, where thought lines of communication between the bars are crossing the zero-line). This critical range begins for metallic hydrogen fractions above 30 per cent with Hmet/LH2 propellant combination and above 5 per cent with Hmet/MA due to increasing chamber temperature.

This means for example that 50% Hmet in a Hmet/LH2 propellant yields a temperature drop of 52.9 per cent above the reference value for the SSME. This can be realized either by 40 per cent increase of the heat flux compared to the SSME with constant overall film coefficient, or by 28.5 per cent decrease of the overall film coefficient with constant heat flux capability. It it obvious, that tremendous efforts in the fields of material research are neccessery yielding material properties capable of meeting those demands.

On the other hand, Fig.3-10 gives the positive result, that today's cooling technology will be sufficient up to 30 per cent of Hmet fraction, representing the potential for low cost chamber technology.

Results:

- The propellant systems Hmet/LH2 and Hmet/MA shows different behaviors concerning cooling requirements.
- Todays cooling technology is applicable up to 30% Hmet for system Hmet/LH2.
- Today's cooling technology is applicable only up to 5% Hmet for the system Hmet/MA.
- These values are optimistic.
- The system Hmet/LH2 offers a great potential for cost savings of chamber and cooling technology if Hmet fractions beneath 40% are chosen due to their low chamber temperatures.
- The system Hmet/MA offers constant cooling conditions in the range between 20% and 60% Hmet weight fraction.
- Enhanced research towards materials with increased thermal conductivity and low thermal expansion coefficients is required.
- Further investigations concerning gas side heat transfer minimization and coolant side heat transfer maximization arenecessary.

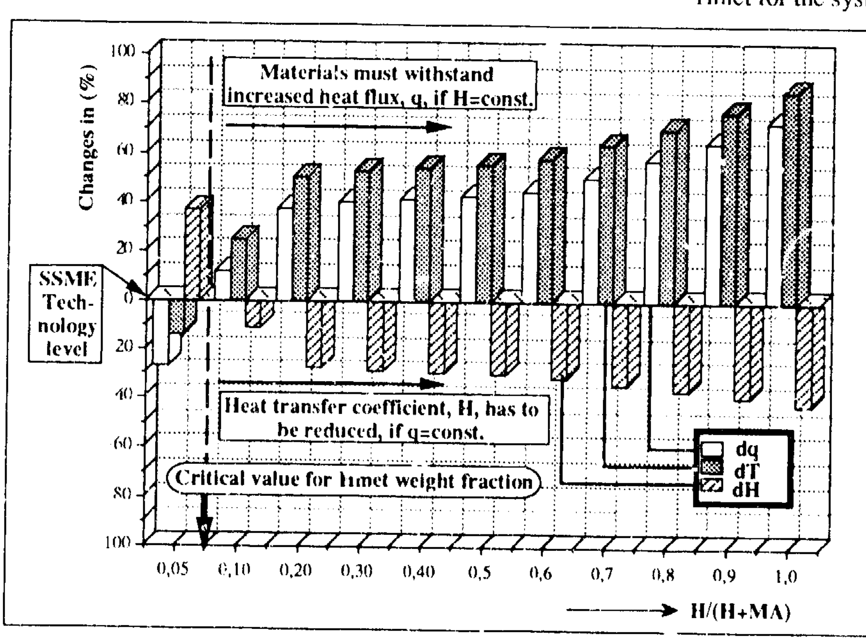


Fig.3-11: Sensitivity of chamber cooling for propellant system Hmet/MA, relative to SSME chamber cooling conditions

3.4 STORAGE CONCEPTS

This chapter contains parametric results of some reflections on basic design configurations of propellant tanks. These investigations base on the assumption of metallic hydrogen being stable at all pressures and in all different states of aggregation.

3.4.1 Influence on propellant volumes

An important propellant parameter is their density. High densities are desirable to minimize the size and weight of propellant tanks and feed systems.
The Hmet weight fraction dependent storage volumes for the propellant combinations Hmet/LH2,SLH2,SH2,RP1,MA have been computed for different payload masses (5 Mg, 20 Mg and 200 Mg) for a 9300 m/s-SSTO-vehicle mission. Absolute values have been connected with corresponding values for a conventional LOX/LH2 system (O/F=5; Expansion ratio 68:1; eq. flow) yielding 4111.67 m/s specific vacuum impulse. In Fig.3-12 the propellant volume fractions are plotted over the Hmet weight fraction for the different propellant combinations.

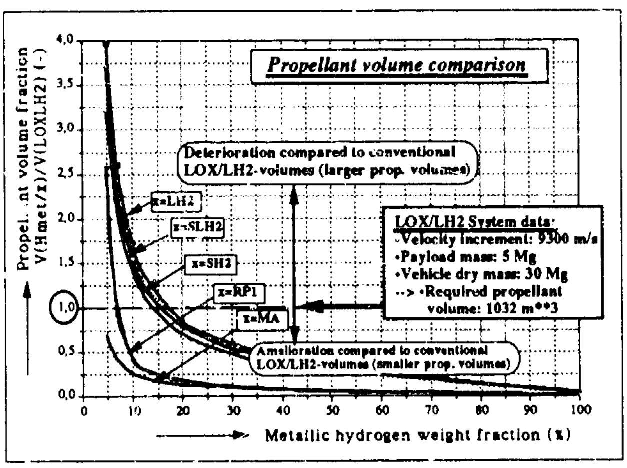


Fig.3-12: Relative changes of propellant volume with metallic hydrogen weight fraction; factors at vertical axis indicate the change to the corresponding values for LOX/LH2 system

All line points above a volume fraction of one are not as good as the conventional LOX/LH2 system, they have larger volumes. All line points belowa volume fraction of one represent potential mass reductions. Hmet weight fractions above 17% yield propellant volume reductions, compared to conventional systems.
In Fig.3-13 the volumes data are expressed in terms of diameters of spherical tanks to increse the vividness.

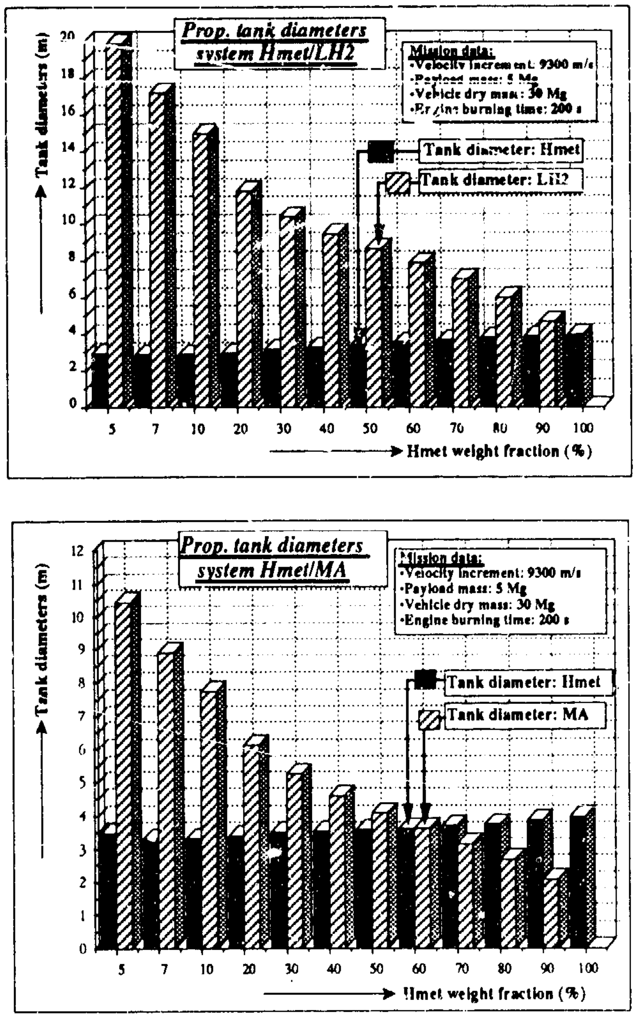


Fig. 3-13: Required volumes expressed as diameters of spherical tanks of propellant components plotted over metallic hydrogen weight fraction

Results:
- If hydrogen in liquid, slush or solid state is used as secondary propellant component at least 15%, 16 respectively 17% mealic hydrogen weight fraction is required to be smaller in propellant tank volume compared to the conventional LOX/LH2 system.
- Most effective reductions in propellant volumes are achieveable with methyl alcohol and RP1 as secondary component. Less than 10% Hmet weight fraction will be sufficient.
- 40% Hmet weight fraction yield propellant volume reductions less than a factor of 0.5 for all systems.
- Increasing Hmet weight fraction reduces secondary component propellant volumes very strongly meanwhile the Hmet propellant volume increases moderately.

3.4.2 INFLUENCE OF PROPELLANT STORAGE TEMPERATURE

Metastable metallic hydrogen propellant could call for storage temperature down to 5 K. If the cryogenic propellant components LH2, SLH2 or SH2 are used, storage temperature still down to 12 K (for SH2) is required. The most serious tank design problems for cryogenic propellants reduces to the design of adequate thermal tank insulation. A simple approximation of the insulation thickness and specific density will answer the question, if storage temperatures in the mentioned ranges will be problematic.
The insulation requirements may be specified for the three phases listed in Tab. 3-5.

Requiring phase	Objective	Material
Ground hold period	To reduce evaporative losses and therewith costs	Insulating blankets (removed prior to liftoff)
Boost phase	To reduce heat transfer due to aerodynamic heating	e.g. Laminated-type insulation
Coast flight in space	To prevent propellant from radiation from sun and the planets	Radiation shield (magnesium oxide, silver applied as coating onto light base aluminum

Tab.3-5: Insulation requirements for propellant tanks

From the mass of propellant tank point of view the ground hold period and boost phase are of importance. To investigate the latter point, the general steady-state heat-transfer equation has been used:

$$q=Q/A=(k/t)\cdot DT$$

where

- q : Heat flux or heat, transferred per unit area per unit time [W/m2]
- Q/A : Heat transferred per unit time [W] across a surface A [m2]
- t : Chamber wall thickness [m]
- k : Thermal conductivity of chamber wall [W/m oK]
- DT : Temp. differential across the insulation (T2-T1) [K]

Tab. 3-6 shows the regarded temperatures during the ground hold phase and the boost phase.

T [K]	Ground hold phase T2g=298			Boost phase T2b=700		
		DT			DT	
T1min	T1gmax	T2g-T1min	T2g-T1gmax	T1bmax	T2b-T1min	T2b-T1bmax
5	12,5	295	285,5	62,5	695	637,5
14	35	284	263	85	686	615
20	50	278	248	100	680	600
77	192,5	221	105,5	242,5	623	457,5
298	298	0	0	348	402	352

T1 : Temperature of insulation surface near the tank wall
T2 : Temperature of the outer insulation surface
g : Ground hold phase
b : Boost phase
T1gmax = T1min•2.5
T1bmax = T1gmax+50 K

Tab. 3-6: Absolute temperatures and temperature differentials

As insulation a honeycomb light weight system has been choosen due to wide application for cryo-tanks. The cross-section through the honeycomb-supported tank structure may be seen in Fig. 3-14. The insulation consists of a honeycomb core filled with foam (isocyanate-type), installed between an inner and outer facing laminate type sheet. The space of the gap between the tank wall and inner insulation surface is purged with helium to reduce the vacuum degradation by infiltration of air and to serve as leak-detection device. This insulation system delivers a thermal conducitivity of about 2.883E-2 W-m/m^2-°K (which is equivalent to 0.2 Btu-in/in^2-sec-°F).

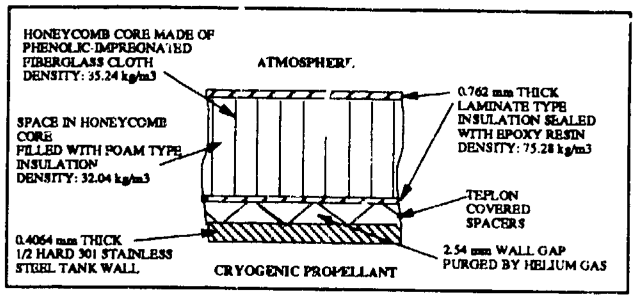


Fig. 3-14: Construction elements of a cryogenic tank insulation design (external type); [from 15]

The relative changes of thickness and density with storage temperature connected to the conventional liquid hydrogen system (T-storage: 20 K) is shown in Fig. 3-15. The approximated values are given for the ground hold and boost phase requirements.

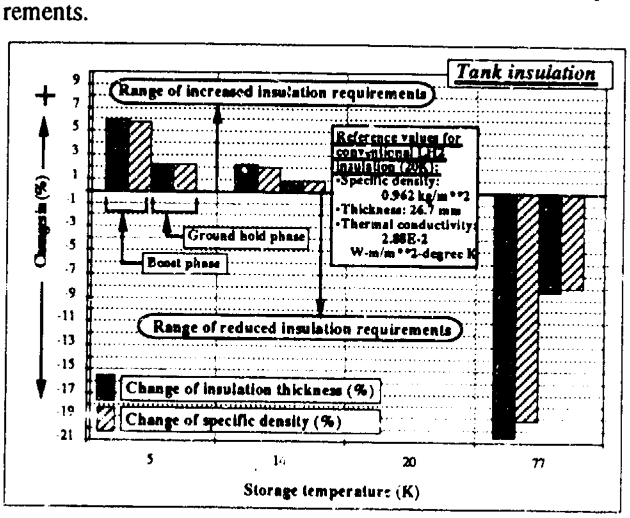


Fig. 3-15: Approximation of per cent changes of the thikkness and specific density of propellant tank insulation material for various propellant storage temperatures, connected to conventional liquid hydrogen storage conditions (20 K)

Results:

- Storage of propellants even at temperatures of about 5 K is technically feasible and will not increase the insulation masses dramatically
- Propellant storage temperatures in the range of slush hydrogen temperature (14 K) respectively solid hydrogen temperature (5 K) require an increase of insulation material thickness of about 2.25%/5.99 % for minimum values of ground- hold phase respectively 0.87%/2.2% for boost phase conditions, compared

to liquid hydrogen storage conditions (T=20 K)

- The increase of required specific density of insulation material will not exceed 5.73% in the case of ground hold phase, respectively 2.21% in the case of boost phase, if storage requires 5 K.
- If storage is needed at 77 K a dramatic decrease of insulation material thickness (-20.6%- ground hold phase) and density (-19.13%- ground hold phase) will happen.

3.5 PROPULSION CONCEPTS

The propulsion concept design will primarily be the question of tank arrangements, dependent on the state of aggregation of the propellants. Although the probability, that metallic hydrogen will exist in a solid state, a liquid system has also been investigated. A general review about the positive and negative characteristics of liquid, hybrid and solid hydrogen propulsion systems will be given below.

3.5.1 Liquid metallic hydrogen propulsion systems (Hmet/LH2,SLH2,RP1,MA)

The liquid systems are the propellant combinations PC1,2,3,4 from Tab.3-2. Fig.3-15.shows the overall propellant tank heights, based on a conventional bipropellant tandem arragement. The maximum diameters have been defined to 6 m for the hydrogen tank and 3.8 m for the MA tank.
As may be seen from Fig.3-1 , the overall tank heights can be reduced dramatically with metallic hydrogen propulsion systems, compared to conventional launchers.
The principles of liquid metallic hydrogen propulsion concepts are shown in Fig.3-17 (p. 12). The feeding of the investigated liquid propellants can be done by conventional methods, either by gas-pressure systems, by turbopump systems or by combinations. A new feeding alternative of Hmet could be realized by electricity due to the conductive properties of Hmet. The required technology could be adopted e.g. from mass driver concepts. Magnetic fields moving down the Hmet propellant ducts acting as mass driver. Disadvantageous is the need of extra power required on board. It is conceivable, that, during the next millenium, energy transmission from the ground to the vehicle e.g. by laser beams could reduce this problem only to one of energy conversion. Some aspects concerning the differnt feeding systems are given:

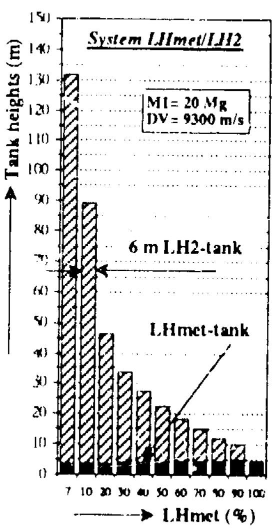


Fig.3-15: Propellant tank dimensions for liquid metallic hydrogen system (m1: 20 Mg; DV: 9300 m/s)

- If metallic hydrogen is available at temperatures in the range of 5 K further research is necessary as to rotating parts of a turbopump system
- Lower mass flows make the use of turbopump system easier
- If lower percentages of Hmet are used (together with over proportional secondary mass requirements) a pressure feed system could be suitable for Hmet-feeding, a turbopump system for the secondary component

3.5.2 Hybrid metallic hydrogen propulsion systems (SHmet-P,G/LH2,SLH2,RP1,MA)

The systems PC5 to PC12 (see Tab.3-2, p. 4) use metallic hydrogen in solid form in conjunction with liquid working fuels. Metallic hydrogen in solid state is most likely. It can be classified as follows:

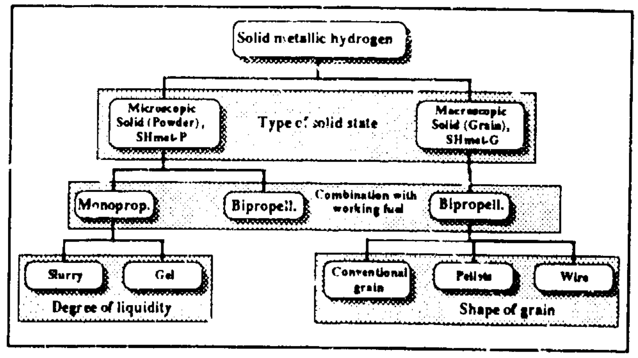


Fig.3-16: Different forms of solid metallic hydrogen

SHmet could be stored as powder (microscopic particle size, SHmet-P) or as grain (macroscopic particle size, SHmet-G). The powder concept can be subdivided according to the number of required tanks into monopropellant and bipropellant systems. Monopropellant systems offer the chance to reduce the complexity of the overall storage system. Solid hydrogen could be used as suspension in direct combination with the secondary propellant component as slurry or gel. The latter is more viscous than a slurry propellant.
The conventional grain concept represents the classical solid grain, which is embedded inside the thrust chamber. Small pellets are stored in an extra tank. They will be inserted into the thrust chamber like the ammunition of a submachine gun. The rolling-up of solid metallic hydrogen wire represent another

grain concept.

Common to all solid concepts except suspensions, is the lower package respectively storage density due to air-spacing. However, this should be without prejudice for the overall system, if the fraction of metallic hydrogen is great enough, as mentioned in previous chapters. The different concepts are explained below. Most of them have been previously described for conventional metal combustion.

3.5.2.1 Powder bipropellant concept

This concept regards the solid hydrogen propellant as powder, which is stored isolated from the respective working fluid. It will further be distinguished between separate injection (concept A) and common injection (concept B) into the thrust chamber, as may be seen from Fig.3-17. The SHmet-P propellant will be fed by high pressure gas. The working fluid will be fed by a turbopump driven by a turbine, based on an open combustion tap-off cycle.

In concept A the SHmet-P will be injected directly into the thrust chamber. Concept B introduce a mixing of the components before injection into the chamber. The mixing process is realized by means of the ejector principle. The main advantages and disadvantages of the two diergol concepts are summarized In Tab.3-7.

<table>
<tr><th></th><th>Concept A</th><th>Concept B</th></tr>
<tr><td rowspan="2">Advantages</td><td colspan="2">•High storage density
•Easy controllable force of reaction</td></tr>
<tr><td>•No mixing of fuels before injection</td><td>•More simple ignition head</td></tr>
<tr><td rowspan="2">Disadvantages</td><td colspan="2">•Feeding of SHmet-P only by means of pressure gas
•Lump appearance in the tank due to humidity
•Dust formation in the tank
•Danger of clogging in the ducts/lines, valves
•Complicated filter technique required
•Complicated tank geometry
•No literature available
•Unfavourable SHmet-P tank form</td></tr>
<tr><td>•Complicated injection system
•Mixing prozess in thrust chamber</td><td>•Design of ejector system
•Temperature gradients during mixing in case of temp. differences</td></tr>
<tr><td>Problems</td><td colspan="2">•Determination of optimal particle size
•Production of homogeneous particle size
•Fueling of SHmet-P
•Flow behavior (wall influences)
•Valve technology
•Monitoring instrumentation (flow, quantities and levels in tank)
•Thermal behavior due to particle friction
•Ignition behavior</td></tr>
</table>

Tab.3-7: Different aspects concerning SHmet-P-bipropellant concepts (A: separate propellant injection; B: common injection)

3.5.2.2 Powder monopropellant concepts (slurry/gel)

A monopropellant system is a potential alternative to reduce the system complexity and therewith a good chance to reduce the space transportation costs. The problems of slurry combustion are described in literature [17]. Although these investigations concentrated on coal-oil slurry combustion, many aspects can be adopted for a metallic hydrogen slurry concept.

Slurry means a suspension of solid particles inside the liquid working fluid.

A gel is a liquid containing a colloidal structural network that forms a continuous matrix and completely encloses the liquid phase. For comparison of both concepts, see Tab.3-8. Illustrations of the concepts can be seen in Fig.3-17.

<table>
<tr><th></th><th>Slurry concept</th><th>Gel concept</th></tr>
<tr><td rowspan="2">Advantages</td><td colspan="2">•Reduction of overall system complexity
•Reduced number of components
•Small vehicle size
•Research projects just under way</td></tr>
<tr><td>•No mixing of fuels before injection</td><td>•No propellant sloshing
•Handling ease</td></tr>
<tr><td rowspan="2">Disadvantages</td><td colspan="2">•Feeding of SHmet propellant by means of pressure gas (if gel propellant)
•Danger of clogging in the ducts/lines, valves
•Complicated filter technique required
•Special pumps necessary due to high viscosity
•Bypass system required (if clogging)
•Flow behavior dependant on temperature
•Coolant of chamber is problematic
•High viscosity (can be reduced by heating)</td></tr>
<tr><td>•Propellant sloshing
•Required constant mixing</td><td>•Storage temperature dependance
•Shear forces required to make it flowing
•Additives (as gellactants) required</td></tr>
<tr><td>Problems</td><td colspan="2">•Filtering of the slurry to prevent clogging
•Mechanical stabilization in the anticipated environment
•Influence of Hmet weight fraction on flow behavior
•Determination of optimal particle size
•Production of homogeneous particle size
•Influence of particle weight distribution
•Fueling
•Flow behavior (wall influences)
•Valve technology
•Monitoring instrumentation (flow, quantities and levels in tank)
•Thermal behavior due to particle friction
•Ignition behavior
•Combustion characteristics (droplet size, evaporation behavior, burning velocity,...)</td></tr>
</table>

Tab.3-8: Comparison of SHmet-P-monopropellant concepts

3.5.2.3 Hybrid grain concepts

Here macroscopic grains of Hmet are regarded. It is completely unknown how solid metallic hydrogen in concentration of 100% will interact with its environment. From solid atomic hydrogen imbedded inside a solid molecular hydrogen matrix, high regression velocities in the range of 2.1 m/s are known. For this study metallic hydrogen mass flows will be assumed to be in the range of today's magnitudes, possibly by means of additives.

In the case of the <u>conventional grain concept</u> solid metallic hydrogen propellant is contained within the combustion chamber, in which the liquid working fluid will be injected. A more detailed critical examination may be seen from Tab.3-9.

The <u>pellet concept</u> means, that spherical pellets of SHmet will be injected into the thrust chamber. Disadvantageous is the lower propellant package density in the tank and fluctuation of the combustion, dependent on the injection rate. Advantageous is the flow rate controllability and engine shut down capability in case of emergency. Production could be easy due to the small size particles, delivered by the diamond anvil cell.

The <u>wire concept</u> is a completely new one. Solid metallic hydrogen is spooled onto a coil which could be driven by electric energy or pressure gas. Full thrust controllability is given. Disadvantageous will be the lower propellant package density combined with additional mass for the coil and bearings

as well as the difficult sealing of the feeding lines. Moreove, the solid propellant has to be pliable.
The different grain concepts are compaired in Tab.3-9 while illustrations are shown in Fig.3-17.

	Conventional grain	Pellets	Wire
Advantages	•High propellant density •Nearest probability of Hmet state		
	•Less cooling problems of the chamber wall •Low package density •Proven concept •Partially thrust control	•Fully thrust control •Feeding problematics known from DAEDALUS •Easy production in anvil cells •Engine shut down	•Fully thrust control •Engine shut down •Mechanical feeding by pressure gas
Disadvantages	•High concentrated energy density		
	•High burning velocity •Additives necessary to reduce burning velocity •Feeding of liquid component by pressure gas	•Package density dependent on pellets size •Sealing •Combustion fluctuation	•Package density dependent on wire geometry •Complicated feeding system •Difficult sealing of feeding lines •Required pliability
Problems	•Influence of electric and magnetic fields •Combustion behavior (velocity, instability)		
	•Combustion cut off	•Optimum pellets size •Statistic deviation of optimum size •Pellets feeding rate	•Propellant distribution in chamber

Tab.3-9: Comparison of potential grain concepts

3.5.3 Solid metallic hydrogen propulsion system (SHmet-P/SH2)

A solid propellant metallic hydrogen propulsion system is conceivable as a solid hydrogen matrix, in which solid metallic hydrogen particles are imbedded. It makes more sense to use a microscopic powder rather than macroscopic solid particlesbecause of the more homogeneous combustion behavior.
A concept like this offers the known conventional advantages, listed in Tab.3-10. They vanish immediately, if e.g. stability of solid particles can't be assured in case of quarrelsomeness between SH2 and SHmet (e.g. due to different component temperatures or mechanical respectively thermal sensitivity). A simple illustration of the concept can be seen in Fig.3-17.

Advantages	•No feed systems •No valves •Simple in construction
Disadvantages	•Difficult propellant production •Low storage temperatures •No thrust modulation •Long duration storage problematic •Cigarette-burner (->small burning area, changing chamber volume)
Problems	•High temperature gradients through propellant during combustion •Unknown combustion reaction velocity as function of % Hmet

Tab.3-11: Analysis of solid propellant metallic hydrogen propulsion concept

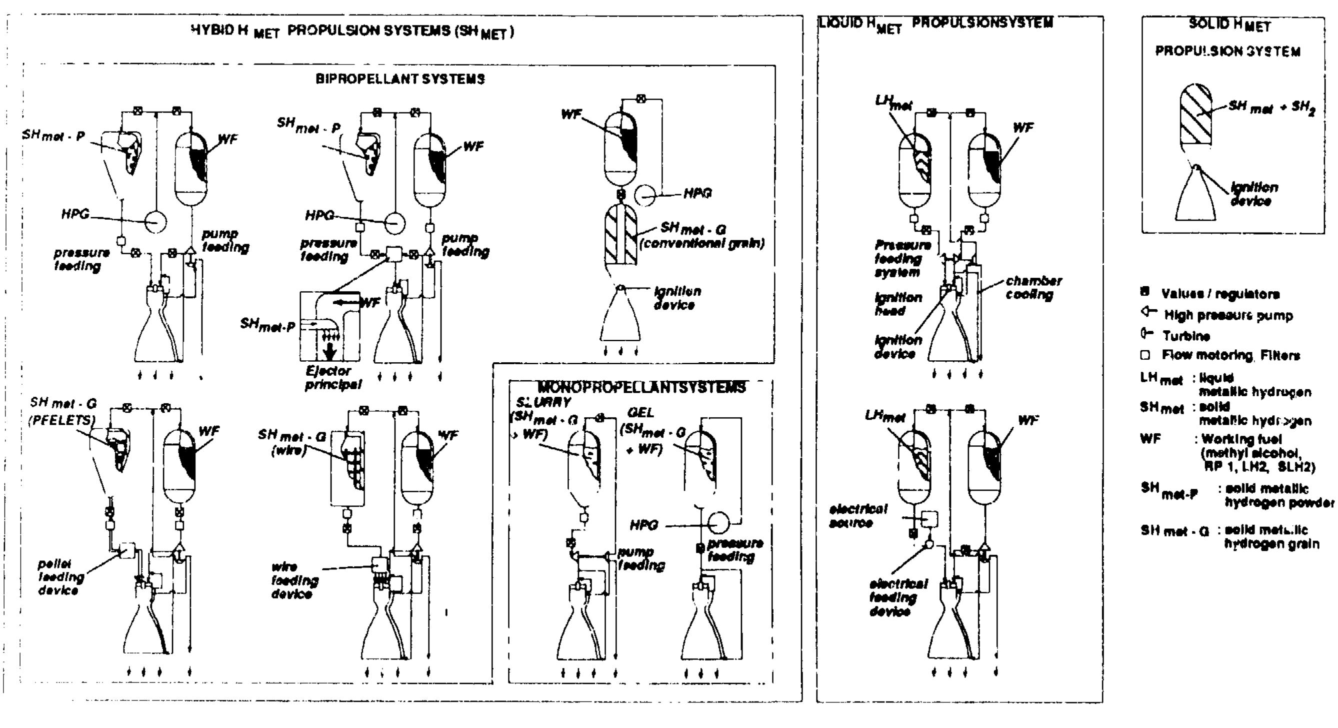


Fig.3-17: Illustrations of possible metallic hydrogen propulsion concepts

3.6 PROPULSION PERFORMANCE

The following figure 3-18 gives the relations between metallic hydrogen percentage in the fuel and global vehicle mass fractions for the most advantageous propellant combination, Hmet/LH2. Of importance arepayload mass ratio M1/M0, dry mass ratio Mn/M0 andpropellant mass ratio M8/M0 which gives total vehicle mass in the sum.

Thevalues given base on simple basic rocket equation calculation rather than on a detailed mass model. Launch masses required for reaching the velocity increment of 9300 m/s with constant vehicle dry mass of 30 Mg, are inserted into the columns for each calculation.

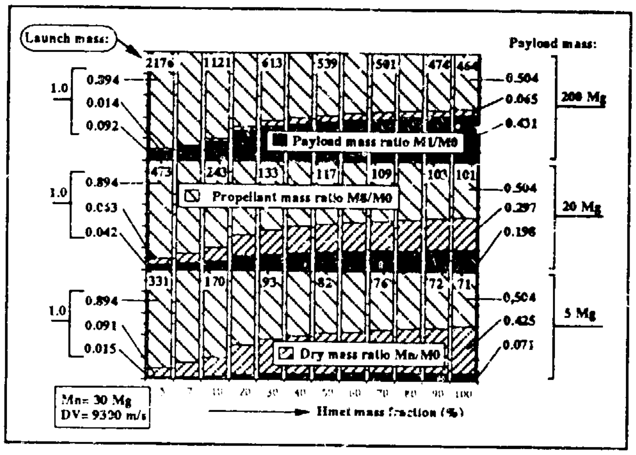


Fig.3-18: Approximations of the main vehicle mass ratios dependent on metallic hydrogen percentage and varying payload masses (SSTO-calculation)

Results:

- Resulting vehicle launch masses for given mission demands are low, compared to existing launchers, even at low Hmet mass fractions.
- Payload mass ratios were kept constant for different payload masses.
- Use of metallic hydrogen propellant yields a great potential for launch mass reductions.
- Increase of dry mass ratio with %Hmet results from constant payload mass.
- At low metallic hydrogen weight fractions of 5%, with a payload mass of 5 Mg represent mass ratios of todays launchers (note:here SSTO calculation with DV=9300 m/s).
- The propellant mass ratio decreases to a minumum of 50.4% with 100% metallic hydrogen .
- Payload mass ratios increase with payload mass for all combinations, while the dry mass ratios decline. Today, dry mass ratios in the range of 10% to 13% are feasible. All computed dry mass ratios above this range (see Fig.1) indicate free mass potentials which could be used either for higher payload masses or for heavier but more stable and therefore saver vehicle structures.
- Dry mass ratios for 200 Mg of payload are lower than 6.5% in each case, which indicates the need for better leight weight structures if payload mass is not to be reduced.

3.7 ENVIRONMENTAL ASPECTS

One dominant characteristics of super high energy metallic hydrogen propellant is the specific impulse. However there exists limits for the specific impulse primarily due to acoustics.The negative effects to the surroundings of the launch vehicle:

- Mechanical effects (e.g. on ground equipment by vibrations etc.)
- Chemical effects (e.g. on the atmosphere the vehicle is flying through)
- Thermal effects (e.g. on the atmosphere or on the ground)
- Acoustical effects (on the surrounding but also on the vehicle itself)
- Emergency destruction effects (around the point of vehicle destruction)

Not all of these effects seem to be critical for the operation of the launch-vehicle. The present paper discusses only the problematic effect, which arises due to the enormous exhaust velocities, dependent on metallic hydrogen weight fraction.

The primary impact comes from the sound, wich is defined as mechanical oscillation inside an acoustic medium. It is measured as sound pressure and sound velocity. The oscillating pressure p has the most dominant destructive influence on technical structures with resulting effects on the environment. Practically the sound pressure level Lp is measured in decibel:

The sound pressure produced by rocket engines can be divided into jet-stream noise and combustion noise. The jet-stream noise rises with the exhaust velocity and is therefore the dominating sound source.

$$Lp\,[\mathrm{dB}] = 10\,\lg_{10}\left(\frac{\tilde{p}^2}{\tilde{p}_0^{\,2}}\right)$$

where:

$\tilde{p}$: Oscillating pressure

To quantify the effect of acoustic noise, the power of sound has been calculated. Between 0.7 and 1.6 the Mach emitted power of sound by a jet-stream is raising with eight to the power of the exhaust velocity v. With an exhaust velocity greater than mach two (v > 2 Mach), the jet-stream power of sound rises with 3 to the power of the exhaust velocity (/27/, p. 281). Fig.3-19 shows the results of the parametric calculation of the acoustic noise level as function of the jet exhaust velocity with the distance r from the source as parameter.

Sound levels over 200 dB are not at all acceptable for technical structures. It should be mentioned that values over 194 dB would be equivalent to an alternating pressure greater than 105 N/m2!

As limiting values for sound pressure level it may be proposed [19]:

-145 dB as maximum stress limit for conventional rocket technical structures

-160 dB as maximum stress for highly stable designed structures and launch facilities (submersible launch towers, concrete shelters for measuring tools etc.).

Whereas from another authority [20] a maximum alternating sound pressure stress of 0.1 bar (= 10^4 N/m^2) may be regarded as tolerable. This limiting value is equivalent to a sound level of 174 dB. Future research activities at the Technical University of Berlin will include detailed investigations of the correlation betwee high exhaust velocities and generated noise power.

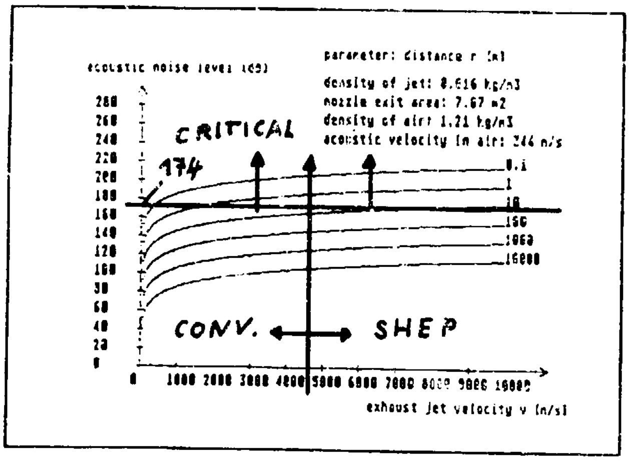


Fig.3-19: Acoustic noise level as a function of exhaust jet velocity with distance r as parameter [19]

Results:

•Gas exhaust velocities above those of conventional systems (4500 m/s), enhance the maximum stresses of structures (not only of vehicle) near destructive limits, due to increased noise levels.

•Metallic hydrogen weight fraction is an important parameter to be considered concerning acoustical effects.

•To take advantage of high energetic propulsion systems, methods concerning effective noise reduction (below 174 dB) have to beinvestigated.

•The only effective method in the moment to reduce the noise power, seems to be air augmentation.

•Vehicle mass savings (due to high propellant energetics) may probably be compensated partly by mechanical devices to realize air augmentation.

3.8 REFLECTIONS ON COSTS

The introduction of a new propulsion concept will be favoured, if there will be a potential for cost savings, namely for development and operation costs. In particular, the specific space transportation costs ($/kg-payload) respectivily the vehicle lauch costs are of importance. They can be reduced through high payload capabilities, high launch rates and low system complexity. Mostly, space system costs can be expressed as function of masses, as have been done in this study. It should be noted, that the following reflections are more general rather than based on detailed analysis.

As may be seen from Fig.3-18 (chapter 3.6), the use of a Hmet/LH2 propulsion sysstem yields overall launch masses, much lower than today's launchers. Hmet vehicles yield furthermore higher payload capabilities with less system complexity. The low launch masses respectively the high launch mass ratios represent a possible potential to reduce the system costs.

In Fig.3-20, the approximated ranges of Hmet-vhicle launch costs are marked, based on a parametric comparison with past, todays and near future launchers.

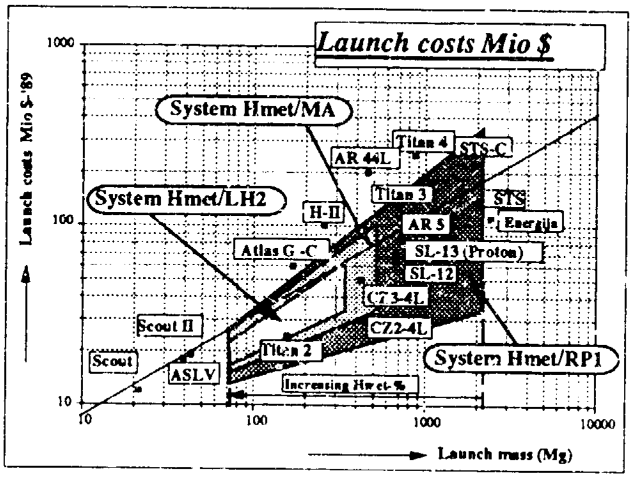


Fig.3-20:Approximation of launch costs of metallic hydrogen transportation systems, compared to various launchers [1]

Conclusions:

•Propellant combinations using more energetic working fluids (first hydrogen, second methyl alcohol, third RP1) yield lower launch masses and hence lower launch costs.

•Launch costs can be reduced with increasing metallic hydrogen weight fraction.

•Launch costs of Hmet-vehicles are much lower than conventional ones carrying the same payload mass.

•Areas for the Hmet-vehicles represent the upper limit of the expected launch costs, because that the mass values are based on SSTO calculations (SSTO systems are less complex compared to staded vehicles, and will therefore lead to reduced launch costs).

Fig.3-21 shows the correlation between payload capability and specific costs ($/kg-payload). The approximated range for Hmet systems is marked. Space transportation based on metallic hydrogen system will be less costly, compared to conventional systems, due to much lower launch masses for the given payload masses.

Further reflections on costs of Hmet-sysems are:

•The super high energetic propellant combination will require enhanced security demands which could lead to higher operational costs.

•Cost reduction potential due to less complex ground infrastructure (smaller propellant storage facilities, smaller hangars and launch towers, etc.)

•Design of reusable space vehicles using metallic hydrogen propellant systems could be advantageous due to smaller overall vehicle size.

•Use of hydrogen based Hmet combination could be much cheaper than hydrocarbon combinations, due to probably spreading hydrogen house keeping world wide during the next millennium.

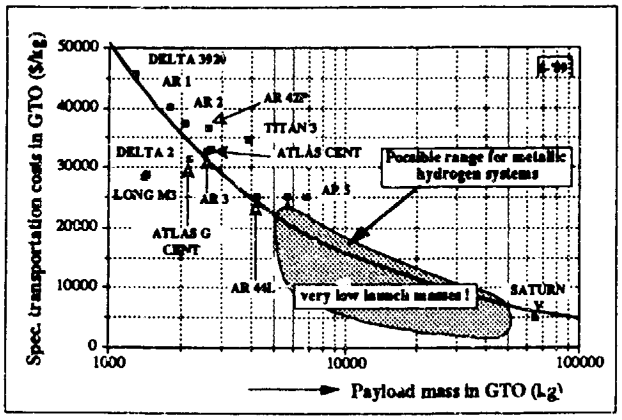


Fig.3-21: Approximation of specific transportation costs ($-'89/kg) of metallic hydrogen transportation systems compared with conventional systems; data of given launchers have been converted into GTO data [1]

4. SUMMARY

The study investigates metallic hydrogen for the application as rocket propellant. Due to its very high theoretical specific energy of 52 kcal/g, yielding a maximum specific impulse of 1700 m/s, and its high density of 1150 kg/m³, metallic hydrogen would be an interesting propellant candidate. However, there are many restrictions concerning the knowledge about metallic hydrogen:

- The required atomic state of Hmet has not yet been proven (required pressures for dissociation in the range between 2.5 and 4.0 Mbar have not been achieved today).
- Uncertainty about the metastability of metallic hydrogen (probably Hmet will not be metastable).
- Uncertainty about basic chemical and physical properties, like state of aggregation (probably Hmet will be a solid).

All investigations took place on condition that Hmet will exist in an atomic state and will be stable.
Different types of potential propulsion systems (liquid, hybrid and solid system) have been analysed. Three different propellant combinations have been compared: metallic hydrogen with hydrogen (liquid, slush and solid hydrogen) as working fluid, with RP1 and with methyl alcohol as working fluid. The main system parameter is the metallic hydrogen mass fraction, which influences the overall propulsion and vehicle performances primarily.
It may be seen from this study that metallic hydrogen is of advantage, compared to propellants used today. The overall vehicle masses enhance the complexity of possible Hmet launchers will decrease, due to increased propulsion performance. Nevertheless, gas exhaust velocities must not be increased unlimited, due to noise impact. For that reason, the Hmet propellant combination using LH2 as working fluid is of advantage, yielding improvements of system performances even with today's exhaust velocities.
The main results and conclusions followed the investigation of combustion characteristics, thrust chamber cooling, storage concepts, propulsion system performance, environmental loads and costs are summarized below:

- All system parameters depend largely on metallic hydrogen mass fraction.
- Combustion gas temperature is mainly dependent on metallic hydrogen mass fraction (up to 6000 K with high percentages; down to 1500 K for low percentages [conventional LOX/LH2 system :3750 K]).
- There are lower thermal risks of thrust chamber due to lower chamber temperatures, if metallic hydrogen propulsion is used delivering specific impulses of today's systems (low percentages of metallic hydrogen).
- An enormous increase in specific impulse arises, if chamber temperatures are not kept within conventional limits (combination using hydrogen as working fluid shows the most extreme behavior).
- The propellant systems Hmet/LH2 and Hmet/MA shows different behaviors concerning cooling requirements.
- Today's cooling technology is applicable up to 30% Hmet for system Hmet/LH2 .
- Today's cooling technology is applicable only up to 5% Hmet for the system Hmet/MA .
- System Hmet/LH2 offers a great potential for cost savings in the fields of chamber and cooling technology.
- Enhenced material research towards increased thermal conductivity and low thermal expansion coefficients is required if high Hmet mass fractions are used.
- Metallic hydrogen weight fractions in the range of 40% will lead to increased payload mass ratios and reduced propellant mass ratios compared to conventional systems.
- Hydrogen as working fluid combined with Hmet represents the most interesting propellant alternative due to
  - lowest overall vehicle masses.
  - highest payload mass potential with respect to realizable dry mass respectively structure mass ratios even at low Hmet weight fractions.
- High percentages of metallic hydrogen may lead to a dramatic decrease of overall propellant volume due to high density of Hmet (1.15 g/cm3).
- The most effective reductions in propellant volumes are achieveable with methyl alcohol and RP1 as working fluids.
- Low temperature propellant storage (for typical prelaunch phases) at about 5 K could be achieved with conventional insulation techniques.
- Metastable liquid state of metallic hydrogen would be of advantage, compared to solid state, due to technical proven components (feeding system, valves, etc.).

- If Hmet was metastable in solid state, a hybrid system would be advantageous, using Hmet as powder or in the shape of macroscopic pellets
- Noise from rocket engines represent one of the most critical impacts to the environment
- The advantage of high specific impulses will be compensated by the disadvantage of enhancement of the maximum stresses of structures near their destructive limits due to acoustic power.
- Methods concerning effective noise reduction (below the limit of 174 dB) have to be researched (air augmentation)
- Vehicle mass savings (due to high propellant energetic and hence low propellant masses) may be compensated partly by mechanical devices required for air augmentation.
- Propellant costs will depend on high pressure facility capabilities
- There is a potential of launch costs and specific transportation costs savings due to the very low vehicle launch masses
- There could exist a cost reduction potential due to less complex ground infrastructure (smaller propellant storage facilities, smaller ground launch facilities, etc.)
- Use of reusable SSTO space vehicles using metallic hydrogen propulsion systems will no longer be a problem

Future research activities should concentrate on the following open technology problem areas:

- Development of high pressure facilities generating required pressures for Hmet production up to 4.0 Mbar
- Detailed mass model, dependent on the storage concept
- Detailed investigation of air augmentation techniques

GENERAL MESSAGE:

- The use of metallic hydrogen as rocket propellant could lead to revolutionary changes in space vehicle philosophy towards low weight, small size and high performance SSTO systems
- Hydrogen (liquid or slush) is the most favourable candidate as working fluid.
- Much more research in the field of high pressure physics is required to come to reliable statements about the chemical and physical properties of metallic hydrogen.
- The technical risks concerning the use of metallic hydrogen as rocket propellant may be controlable.

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